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RESEARCH MEMORANDUM

PERFORMANCE OF AIR INLETS AT TRANSONIC AND LOW SUPERSONIC SPEEDS

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NATIONAL ADVISORY COMMITTEE FOR AERONAUTICS

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The purpose of the present paper is to discuss the transonic air inlet problem and to summarize pertinent information obtained recently. A few introductory remarks are made first in order to indicate the relationship of the transonic problem to the supersonic and subsonic problems.

The primary objective in the design of any air inlet is, of course, the attainment of high internal-flow pressure recovery and low external drag. In the low-speed case the main problem involved in the design of the familiar types shown in figure 1 is that of avoiding flow separation. The broken lines (long and two short dashes) shown define the basic bodies within which the inlets are assumed to be installed. The design of the scoop and wing inlets is somewhat more difficult than that of the nose inlet. In the case of the scoop inlet the initial boundary layer, which exists ahead of the entrance, usually requires special handling to avoid important losses in pressure recovery. In the case of the wing inlet, the angle-of-attack problem is more severe than that for the other types, and special attention must be paid to avoiding adverse effects of the inlet on the lift characteristics of the wing. In general, however, all three types can be designed so that high-pressure recovery is obtained and so that the inlet body will have an external drag as low or lower than that of the basic body as defined by the broken lines. As a result, the choice of inlet type usually is determined by the designer on the basis of other considerations.

In the supersonic speed range a pressure drag exists for a body even though the flow about the body is smooth and unseparated. One of the principal objectives in the design of the supersonic inlets, such as those shown in figure 2, is, therefore, that of minimizing this pressure drag. In general, this objective is attained by using lips sharp enough to permit early shock attachment and by keeping the slopes of the external surfaces as low as possible at all points. The broken lines shown again define the basic bodies to which the inlets are assumed to be applied. In the case of the wing inlet, the simple type shown on the left is formed by merely splitting the basic airfoil along its chord line and separating the halves. Lower drag can be obtained at the higher Mach numbers, of course, by going to the type of wing inlet design shown at the right.

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Another difference from the subsonic case is that the main part of the losses in internal-flow pressure recovery come about through shock losses. Most of the supersonic inlets, therefore, incorporate special means such as protruding central bodies or internal contractions in order to accomplish efficiently the supersonic part of the compression of the entering flow.

The performance characteristics of the supersonic inlets are similar in some respects to those for the subsonic inlets. For one thing, all the types shown in figure 2 can again be installed in the basic bodies defined by the broken lines with little, if any, increase in drag. The design of the scoop and wing inlets is again complicated by fuselage boundary layer and angle-of-attack effects, respectively.

The transonic range is a transition zone for inlets as it is for wings. As the flight speed is increased into the high subsonic range, it becomes necessary to design the subsonic inlets shown in figure 1 for increasingly lower induced surface velocities in order to delay the compressibility drag rise and, in the case of the scoop inlet, to avoid large losses in pressure recovery due to shock-induced flow separation ahead of the entrance. As the speed is increased in the transonic and supersonic ranges, inlets of this type can be made to work satisfactorily by going to higher and higher fineness ratios and sharper and sharper lips. Eventually the optimum geometry becomes that of the supersonic inlet. Transonic inlets, then, are not a new class of inlet but are related to the basic subsonic and supersonic types. The performance characteristics of both types must therefore be evaluated in the transonic range.

One very important problem encountered by both the transonic and supersonic inlets, the so-called inlet-engine air flow matching problem, deserves special mention. This problem arises because the sizing of these inlets is much more critical than the sizing of the subsonic inlet.

Consider in figure 3 the case of a supersonic inlet operating in an off-design condition well below its shock-attachment Mach number and supplying air to a turbojet engine operating at a given rotational speed. Since the turbojet engine is essentially a constant quantity machine when operating at a fixed rotational speed, the engine inlet velocity V_2 has a fixed value. In the optimum case, shown at the top of the chart, the inlet size is such that the normal shock is located just inside the entrance. Inasmuch as the pressure losses across this shock are small, the over-all pressure recovery and, consequently, the mass flow corresponding to the engine inlet velocity are a maximum. If, as illustrated in the middle sketch, a smaller inlet is used, the inlet will deliver insufficient air flow to the engine. The correct volume flow at the compressor inlet is obtained through the mechanism of a decrease in flow density brought about by an increase in the pressure losses across the normal shock which is sucked well down the diffuser. If, as illustrated in the bottom sketch, a larger inlet is used, the normal shock must occur ahead of the lip in order to spill the excess flow around the entrance. This spillage results in a large increase in drag.

The matching problem arises because the optimum inlet size just discussed varies with speed and altitude. In other words, a supersonic inlet sized for optimum performance at one flight condition may be far from optimum with regard to pressure recovery or drag at other flight conditions. In most cases, variable inlet geometry schemes are necessary in order to obtain acceptable performance in the supersonic range. The problem is also of great importance in the transonic range. In almost every case in the transonic range, a turbojet inlet designed for optimum performance at supersonic speeds will be much too small so that the inlet will choke and important losses in pressure recovery will come about through internal shock losses. Some variable-geometry scheme is therefore also vitally needed in the transonic range in order to provide an increase in inlet area large enough to avoid these effects.

From this introductory discussion, it is evident that there are two principal objectives of transonic air-inlet research. One is to learn how to design satisfactory inlets for transonic airplanes. The other is to determine the transonic performance of the supersonic inlets and, where necessary, to learn how to improve this performance to an acceptable level. This research necessarily involves detailed consideration of the inlet-engine air-flow matching problem and of associated problems introduced by the use of variable inlet geometry. With regard to the essentially transonic inlets, the characteristics of the simple open-nose type are discussed first.

Drag results determined by the rocket-model technique for a parabolic-arc body equipped with a pointed solid nose and an NACA 1-40-250 subsonic-type nose inlet (reference 1) are shown in figure 4. (Symbols are defined in the appendix.) It will be recalled that the second and third groups of numbers in this designation show, respectively, that the inlet has a throat diameter equal to 40 percent of the maximum body diameter and a forebody length equal to 250 percent of the maximum body diameter. It will be noted that the drag coefficient of the basic body was reasonably low at supersonic speeds when the drag of the fins is considered. The drag coefficient of the inlet body at the maximum mass-flow ratio was lower than that of the basic body up to some low supersonic Mach number, roughly between 1.1 and 1.2. Above this point, the drag coefficient of the inlet body continued to increase slowly with increasing Mach number and became much larger than that of the basic body at the higher Mach numbers. This result is not necessarily characteristic for the open-nose inlet. The results of reference 2 and other results to be presented subsequently show that drag coefficients closely approaching those for the basic body can be obtained at the higher speeds by increasing the fineness ratio of the inlet and sharpening its lips.

Another point of interest in figure 4 is that, in the lower part of the supersonic range, the drag coefficient of the inlet body increased slowly at first as the mass-flow ratio was decreased below the maximum test value. At design Mach numbers up to 1.4 or 1.5, it is possible to avoid choking at the lower speeds by sizing the entrance for a mass-flow ratio only a small amount (0.1 to 0.3) less than the maximum possible value. It therefore appears that the use of variable inlet geometry can be avoided with this type of inlet at comparatively small cost in drag in the low-supersonic-design-speed case by simply choosing an entrance area slightly larger than the minimum required in the design condition. The results of reference 2 and other results to be presented subsequently indicate that this conclusion is applicable to open-nose inlets with very much sharper lips than the one shown.

A number of NACA 1-series nose inlets have been investigated in the Langley 8-foot transonic tunnel at Mach numbers from 0.6 to about 1.1. As indicated in figure 4, all or most of the transonic drag rise usually occurs below the upper limit of this range. Figure 5 presents preliminary drag results for zero angle of attack and a mass-flow ratio of 0.95 expressed in terms of the increment in external drag coefficient caused by replacing the solid nose of the basic body shown at the top of the chart with the inlet nose. The top group of curves shows some effect of varying the proportions of the nose inlet. The external drag increments due to installation of the two shorter inlets, which had entrance diameters of 40 and 50 percent of the maximum body diameter, were small or negative in the subsonic range. In the transonic range, these inlets increased the drag by maximum increments of 20 to 30 percent of the drag coefficient of the basic body. The third inlet, which had the same entrance diameter as the first but twice its length, did not cause any incremental drag increase in the transonic range up to the maximum test Mach number. This result emphasizes the need for using a very high-fineness-ratio inlet in the transonic range.

The bottom group of curves presents drag increments for a short-nose inlet with twice the entrance diameter of the first inlet and equipped with central bodies possibly suitable for propeller spinners or radar installations. The elliptical-nose configuration had appreciably lower drag than the one with the conical nose. Other investigations (references 3, 4, and 5) indicate that this difference in drag probably is associated with differences in flow angle at the inlet lip. The drag of the elliptical-nose configuration also was as low or lower than that of the two short open-nose inlets of the top group, which had approximately the same induced velocities. This result indicates that properly designed central bodies can be added to inlets of this type at little cost in drag.

An investigation is being conducted currently by means of the rocket-model technique to study the effects of lip shape and inlet profile on the external drag characteristics of open-nose inlets in the low supersonic range. The test vehicle is shown in figure 6 together with the three inlet configurations that have been studied so far. These three inlets differed in exterior inlet profile, but all had the same inlet diameter and forebody length. The bluntest inlet had the exterior profile of an NACA 1-49-300 nose inlet. The inlet of intermediate profile had an exterior lip angle of 9.5° with respect to the body axis and a parabolic-arc transition fairing from the lip to the maximum-thickness station of the body (where the axis of the parabola was located). The sharpest inlet had a conical exterior surface from the entrance to the maximum-diameter station and an exterior lip angle of only 4.9°.

One feature of the test technique requires special mention. The desired internal mass flow in the supersonic range was obtained by providing a sonic-throat (choking) station of the proper size in the internal ducting. A range of internal-mass-flow ratios for a given external profile was obtained by flying separate models with this external profile but with different choking areas. As indicated in the blown-up view of figure 6, the choking station was located just inside the entrance so that changes in internal mass flow were accompanied by changes in internal contraction (internal-lip fairing shape) just inside the entrance. The effects of these changes in internal lip shape on the external drag characteristics of the model are believed to be insignificant. As indicated at the right of the sketch of the test vehicle, the tail cone was lengthened as the choking area was reduced in order to keep the exit velocity approximately independent of the changes in internal mass flow. Because of the change in throat area just inside the entrance, the mass-flow ratio used in presenting the drag data for this investigation is based on the inlet capture area rather than on the inlet throat area as in the rest of this paper.

Drag data obtained to date for Mach numbers of 1.2 and 1.4 are presented in figure 6. Two important conclusions concerning the effects of inlet profile are indicated. First, as previously mentioned, the external drag coefficient of the inlet body for the high mass-flow ratio conditions is reduced importantly, even at these low supersonic Mach numbers, by sharpening the inlet lips and reducing the over-all bluntness of the external lip profile. Second, by sufficiently sharpening the inlet profile, the external drag coefficient of the inlet body can be made to decrease with increasing Mach number in this range rather than to increase as is characteristic for inlet bodies with relatively blunt lips. (See fig. 4.)

Another very important point shown by these data (fig. 6) was referred to previously in connection with the discussion of figure 4: the external drag coefficient of the model of intermediate lip profile increased with decreasing mass-flow ratio at only a slightly greater rate than that for the NACA 1-series inlet which had a well-rounded external lip fairing. This rate of increase in external drag coefficient was much smaller than the rate of increase of the calculated additive drag coefficient which is shown at the bottom of figure 6. Most of the increase in additive drag apparently was compensated for by a decrease in the pressure drag of the external surface of the body. Thus, it appears that the use of a sharp inlet lip does not necessarily preclude the use of a design mass-flow ratio low enough to avoid the necessity for variable inlet geometry in the case of an open-nose inlet designed for low supersonic speeds.

Pressure-recovery results for two open-nose inlets are presented in figure 7. The inlet on the left, which was investigated at 0° angle of attack by the rocket-model technique (reference 2), had only a small amount of internal-lip rounding and an initial conical diffuser angle of only $2\frac{1}{2}^{\circ}$. The internal area-expansion ratio between the inlet throat and the end of the diffuser just ahead of measuring station 2 was 2.3 to 1.0. The variation of pressure recovery with mass-flow ratio for this inlet was fairly flat over the entire test range of Mach number from the lowest test values of mass-flow ratio to the choking values which correspond to the abrupt downward breaks at the right ends of the curves.

The inlet on the right, which was investigated in the Langley 8-foot transonic tunnel, had a much more pronounced rounding of the inner-lip fairing than the other inlet and a diffuser area-expansion ratio of 4.1 to 1.0. At an angle of attack of 0°, the pressure recovery for this inlet began to decrease at mass-flow ratios appreciably below the ultimate choking values at Mach numbers of both 0.6 and 1.1. Surface-pressure measurements show that this decrease was associated with the formation of local regions of supersonic flow on the inner-lip fairing terminated by normal shocks. Thus, differences in inner-lip fairing shape as well as differences in diffuser geometry may have contributed to the markedly different internal characteristics of these two inlets in the mass-flow-ratio range just below choking. Increasing the angle of attack from 0° to 10° caused an appreciable reduction in the mass-flow ratio corresponding to the knee of the curve of pressure recovery plotted against mass-flow ratio at $M_0 = 0.6$ but had a much smaller effect at $M_0 = 1.1$.

Pressure-recovery results for these two inlets for an angle of attack of 0° and a possible design mass-flow ratio of 0.7 are cross-plotted as a function of Mach number in figure 8. It will be noted that the pressure recoveries of both inlets were in the vicinity of 99 percent at subsonic speeds and the pressure recovery of the inlet in the parabolic body closely approached the pressure recovery across a normal shock at supersonic speeds. Maximum pressure recoveries for two conical-shock supersonic inlets measured after diffusion of the internal flow to very low velocities (reference 6) are shown in the figure to permit a comparison. It is seen that the curves for the 30° and the 25° semi-conical-angle supersonic inlets cross the curve for the open-nose inlet at Mach numbers of about 1.4 and 1.5, respectively. This result roughly indicates the range in which the advantage with respect to pressure recovery shifts from the one type to the other.

The fuselage scoop becomes of interest when the nose of the fuselage is needed for various types of equipment. The forward underslung type has several distinct advantages. First, the boundary layer is very thin so that special means for boundary-layer control may not be required. Second, angle-of-attack effects tend to be favorable. Third, at supersonic speeds, an inlet so located can take advantage of the flow compression afforded by the nose shock of the body.

The forward underslung scoop shown in figure 9 was investigated in the Langley 8-foot transonic tunnel at Mach numbers ranging from 0.6 to 1.1. This inlet had rounded lips incorporating NACA 1-series nose-inlet ordinates, a throat area equal to 16.7 percent of the frontal area, and an area-expansion ratio of 2.3 to 1.0 between the inlet throat and the diffuser-measuring station. The entrance was made roughly elliptical in shape in order to obtain a large capture area without increasing the frontal area of the assumed basic body, which again is identified by the broken lines.

As shown in the left part of figure 9, the inlet afforded a pressure recovery of 96 percent or better at Mach numbers of 0.6 to 1.1 at all mass-flow ratios below the choking values. Increasing the angle of attack from 0° to 10° had negligible effect on the pressure recovery and choking values of mass-flow ratio at both Mach numbers.

Drag results are shown in the right part of figure 9 in terms of the increment in external-drag coefficient caused by installing the inlet in the basic body defined by the broken lines. At a mass-flow ratio of 1.0, the drag increments due to the inlet were small or negative throughout the test Mach number range for an angle of attack of 0°.

Increasing the angle of attack to 10° generally decreased the drag increments due to inlet installation. When the mass-flow ratio was reduced to 0.6, the drag increments became positive at both angles of attack. These increments for the mass-flow ratio of 0.6 range in magnitude from 2 to about 35 percent of the drag coefficient of the basic body, depending on the Mach number and angle of attack. It should be pointed out that a mass-flow ratio of 0.6 is well below the values usually encountered in this Mach number range for a turbojet inlet of this type designed for low supersonic Mach numbers.

The pressure-recovery characteristics of the three forward underslung scoops shown in figure 10 have been studied at Mach numbers slightly above 1.4. The configuration shown at the top of figure 10 had a sharp-edge circular entrance located slightly below the fuselage contour and a thin bell-mouth inner-lip fairing. This inlet was investigated in conjunction with a basic pointed fuselage nose (A) and with two alternate spherical fuselage noses (B and C). With the basic pointed nose A, the pressure recovery at the end of the 3.1 to 1 area-ratio internal diffuser was greater than the normal-shock value throughout the entire test range of mass-flow ratio. Replacing the pointed nose with spherical noses B and C caused only small losses, 2 to 4 percent, in pressure recovery.

The scoop inlet in the center of the figure was similar to the inlet discussed in figure 9 except that the inlet lips were sharp. The pressure recovery measured for this scoop after an internal area-expansion ratio of 1.5 to 1 again was greater than the normal-shock value over a wide range of mass-flow ratio below the choking value. The pressure recovery was slightly lower than that for the inlet just discussed, however, because more of the fuselage boundary layer was taken in. Increasing the angle of attack increased the pressure recovery by increasing the amount of flow compression afforded by the nose shock of the fuselage and by causing some of the boundary layer at the bottom of the fuselage ahead of the entrance to flow upward around the sides of the fuselage nose and thus to bypass the entrance.

The inlet at the bottom of the figure was identical to the one just discussed except that the entrance was sweptback. The pressure recovery obtained was lower than that of the unswept inlet except at the highest mass-flow ratios at an angle of attack of 10°. Shadowgraph observations and surface pressure measurements showed that the swept sidewalls were responsible for this decrease in pressure recovery. The normal shock, which occurred ahead of the bottom section of the inlet lip at all mass-flow ratios, caused a very appreciable thickening of the fuselage boundary layer. The swept sidewalls of the inlet confined this boundary layer and forced it to enter rather than to flow sideways and bypass the entrance as occurred in the case of the unswept scoop.

The results given in figures 9 and 10 indicate that the forward underslung scoop can provide performance approximately equal to that for the nose inlet. A pressure recovery higher than that for the open-nose inlet can be obtained at those supersonic speeds for which the normal-shock loss becomes appreciable. As the scoop is located farther rearward along the fuselage, the design problem becomes more difficult because the initial boundary layer becomes thicker and because the local velocities in the region of the inlet usually are higher than those for the forward underslung scoop. Transonic investigations of a number of rearward-located scoops are under way currently but have not yet progressed far enough to provide significant new data.

Another configuration of considerable interest currently is the wing-root inlet. Low-speed results for the inlet illustrated in figure 11 have been published in reference 7. This inlet has now been investigated in the Langley transonic blowdown tunnel at Mach numbers up to 1.4. The wing of the basic model, which again is defined by the broken lines, was composed of 8-percent-thick sections streamwise and had 47° of leading-edge sweep and 0.6 taper ratio. In order to permit installation of the inlet, the wing was flared from the original section at the outboard end of the inlet to a 13-percent-thick section of twice the original chord at the fuselage. The inlet lips were then faired in as shown in section AA by using existing wing-inlet section data as a guide. The entrance throat area was 17.2 percent of the fuselage frontal area. The blown-up view shows a boundary-layer bypass scoop which was studied in the course of the investigation. The arrow shows the flow entering the bypass and then leaving the model through an exit at the bottom of the wing. The pressure recovery was measured after 4-percent internal area expansion at the station where the ducts join. The tail section of an actual airplane fuselage would extend much farther rearward than the model fuselage. However, in the case of some fighter installations, it would still be necessary to use more abrupt bends than the ones tested in order to obtain room for the engine. On the other hand, if the airplane was large and the engines were submerged in the wings as in the case of the Vickers Valiant, no S-shaped bends would be required.

As illustrated by the results for a Mach number of 1.0 shown in the top left part of figure 12, the pressure recovery of this wing-root inlet was affected only a small amount by variations in mass-flow ratio and angle of attack over the ranges investigated. The dashed part of these curves defines the region in which twin-duct instability was encountered. Pressure recoveries for a possible design mass-flow ratio of 0.7 and an intermediate angle of attack of 4.4° are presented in the top right part of figure 12 as a function of the free-stream Mach number. As shown by the solid line, the pressure recovery of the model without the boundary-layer bypass scoops was 90 percent or greater up to a Mach number of about 1.28. The circular symbols show the recoveries obtained with the boundary-layer scoops installed and bypassing a flow quantity equal to about 8 percent of the flow through the main ducts. Installation of these bypasses increased the pressure recovery by $2\frac{1}{2}$ percent at a Mach number of 1.28 without increasing the drag appreciably and provided a pressure recovery of nearly 88 percent at a Mach number of 1.4. This recovery is regarded as satisfactory in view of the fact that the flow has passed through the nose shock of the fuselage, a normal shock ahead of the entrance, and the S-shaped bends.

Drag coefficients based on the wing area of the basic model are given at the bottom of figure 12. The results at the left show that the drag increased slowly with decreasing mass-flow ratio after the fashion of the open-nose inlets discussed previously. The curves at the right compare the drag coefficients of the inlet model without the boundary-layer bypass at the possible design mass-flow ratio of 0.7 with the drag coefficients of the basic model. The drag coefficients of the inlet model were greater than those for the basic model over most of the Mach number range. The maximum positive increments in drag coefficients shown occur in the transonic range and vary from about 5 percent of the corresponding drag coefficient of the basic model at an angle of attack of 0.4° to about 9 percent at an angle of attack of 8.1°. It should be noted in figure 11, however, that installation of the inlet increased the exposed wing area of the model by about 7 percent. If this increase in wing area is taken into account in the comparison, it can be seen that installation of the inlet was accomplished at very small cost in drag. The circular symbols are again for the case in which the boundary-layer bypass scoops were installed. These results show, as previously noted, that the drag increase due to installation of the bypass was negligible at the higher Mach numbers. These preliminary pressure-recovery and drag data show that the swept-wing root inlet is a very promising configuration for use in the transonic range when the inlet must be located well back of the fuselage nose.

Several investigations of the performance of sharp-edge supersonic inlets at transonic and subsonic speeds are currently under way. Because of the nature of the inlet-engine air-flow matching problem, the emphasis in these investigations has been placed on the study of inlet performance at high and choking mass-flow ratios. Figure 13 presents some preliminary pressure-recovery results from an investigation of the performance of a sharp-edge supersonic inlet at transonic and supersonic speeds. These results were obtained in the Langley 8-foot transonic tunnel for a model with a conical-shock nose inlet designed for a Mach number of 2.0. The pressure recovery was measured at station 2 after an internal-area expansion ratio of 1.5 to 1.0. The pressure recovery at an angle of attack of 0° was above 98 percent over a broad range of flow rates at all the test

Mach numbers which ranged from 0.6 to 1.1. The main effect of increasing the angle of attack from 0° to 10° was to decrease pressure recovery at the higher mass-flow ratios and to decrease the choking value of mass-flow ratio at $M_0 = 1.1$ a small amount.

Drag measurements were obtained during the tests, but the data have not yet been reduced to usable form. Schlieren photographs indicate, however, that external-flow separation from the inlet lips did not occur over the important range of operating conditions (at high values of mass-flow ratio) except in the form of localized bubbles. This result is in agreement with the results of the previous low-speed tests reported in reference 8. It is therefore indicated that the sharp lips of the supersonic-type inlets are not necessarily responsible in themselves for excessively large losses in pressure recovery or increases in drag in the transonic range.

Rounding the lip of the supersonic-type inlet has been proposed frequently as a means for improving the mass-flow and pressure-recovery characteristics of this type of inlet in the subsonic and transonic regimes. The pressure-recovery characteristics of the conical-shock inlet shown in figure 14 were investigated with a facility of the Gas Dynamics Branch at an angle of attack of 0° with the original sharp lip, with a thick round lip, and with an NACA 1-series nose-inlet lip of intermediate thickness and rounding. The pressure recovery was measured at a station well downstream of the region of the diffuser shown in the sketch, at which station the velocities were reduced to very low values. At a Mach number of 0.10, which is of interest for take-off, the choking mass-flow ratio was much higher with the two round lips than with the sharp lip. Both of these lips also provided much higher pressure recoveries than the sharp lip at mass-flow ratios greater than 2.0. At a Mach number of 0.80, the NACA 1-series lip provided the highest mass-flow ratio but was only slightly better than the sharp lip. At the Mach number of 1.3, the sharp-lip inlet was superior to both of the round-lipped inlets with respect to both mass-flow ratio and pressure recovery. These results and those of the preceding figure indicate that moderate rounding of the lip of the supersonic inlet, although not effective at transonic speeds, can provide significant improvements in inlet performance in the subsonic range. Any such improvement must, of course, be weighed against the cost in drag at supersonic speeds due to rounding the lip.

CONCLUDING REMARKS

In conclusion, it has been shown that inlets with acceptable performance in the transonic range can be designed by the use of information currently available. More work is needed to define optimum configurations and to establish procedures for their selection and detailed design. In the case of the supersonic inlet, preliminary results obtained so far indicate that the sharp lips of these inlets do not necessarily cause excessively adverse effects on pressure recovery, mass flow, or drag. It therefore appears that the emphasis in transonic research on this type of inlet should be placed on the study of the various variable-geometry schemes which have been proposed in connection with the inlet-engine air-flow matching problem.

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APPENDIX

SYMBOLS

A	area
C_{D_E}	drag coefficient of basic body corrected to free-stream base pressure $\left(\frac{D + A_j(p_j - p_0)}{q_0 A_m}\right)$ or external-drag coefficient of inlet body $\left(\frac{D - \left[m(V_0 - V_j) - A_j(p_j - p_0)\right]}{q_0 A_m}\right)$
ΔC_{D_E}	external drag coefficient of inlet body minus drag coefficient of basic body
D	total drag
m/m_0'	mass-flow ratio based on minimum (throat) area of entrance
m/m_0	mass-flow ratio based on capture area (area bounded by locus of lip leading-edge points)
m	rate of internal mass flow
m_0'	rate of mass flow in free stream through stream tube with area equal to minimum (throat) area of entrance
m_0	rate of mass flow in free stream through stream tube with area equal to capture area
M	Mach number
p	static pressure
P	total pressure
R	Reynolds number based on maximum body diameter
V	velocity
α	angle of attack

θ_c cone semiangle

Subscripts:

0 free stream

2 measuring station at end of internal diffuser

j exit or base area

m maximum area station of body

REFERENCES

1. Sears, Richard I., and Merlet, C. F.: Flight Determination of the Drag and Pressure Recovery of an NACA 1-40-250 Nose Inlet at Mach Numbers from 0.9 to 1.8. NACA RM L50L18, 1951.

2. Sears, Richard I., and Merlet, C. F.: Flight Determination of Drag and Pressure Recovery of a Nose Inlet of Parabolic Profile at Mach Numbers from 0.8 to 1.7. NACA RM L51E02, 1951.

3. Nichols, Mark R., and Keith, Arvid L., Jr.: Investigation of a Systematic Group of NACA 1-Series Cowlings with and without Spinners. NACA Rep. 950, 1949. (Formerly NACA RM L8A15.)

4. Pendley, Robert E., and Robinson, Harold L.: An Investigation of Several NACA 1-Series Nose Inlets with and without Protruding Central Bodies at High-Subsonic Mach Numbers and at a Mach Number of 1.2. NACA RM L9L23a, 1950.

5. Pendley, Robert E., Robinson, Harold L., and Williams, Claude V.: An Investigation of Three Transonic Fuselage Air Inlets at Mach Numbers from 0.4 to 0.94 and at a Mach Number of 1.19. NACA RM L50H24, 1950.

6. Ferri, Antonio, and Nucci, Louis M.: Preliminary Investigation of a New Type of Supersonic Inlet. NACA TN 2286, 1951.

7. Keith, Arvid L., Jr., and Schiff, Jack: Low-Speed Wind-Tunnel Investigation of a Triangular Sweptback Air Inlet in the Root of a 45° Sweptback Wing. NACA RM L50I01, 1950.

8. Dennard, John S.: An Investigation of the Low-Speed Characteristics of Two Sharp-Edge Supersonic Inlets Designed for Essentially External Supersonic Compression. NACA RM L7D03, 1947.

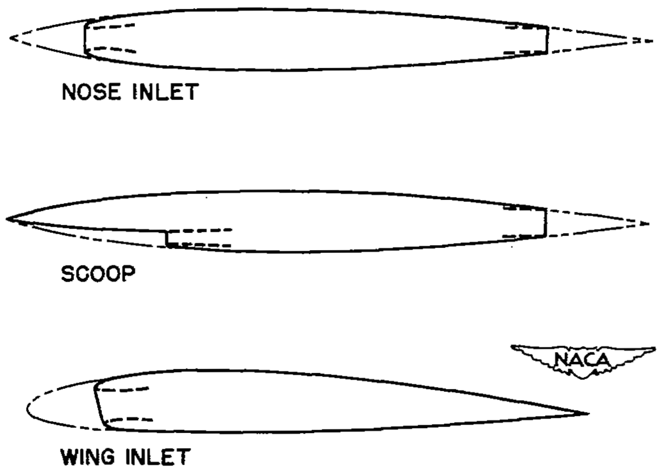

Figure 1.- Subsonic air inlets.

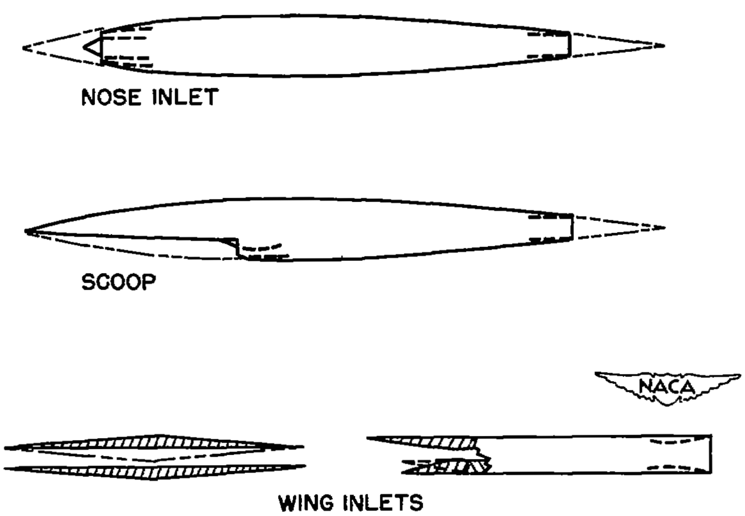

Figure 2.- Supersonic air inlets.

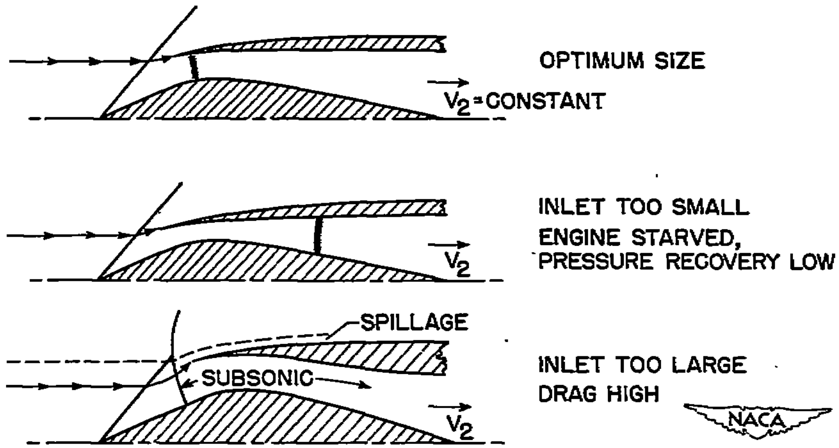

Figure 3.- Effect of sizing on inlet performance.

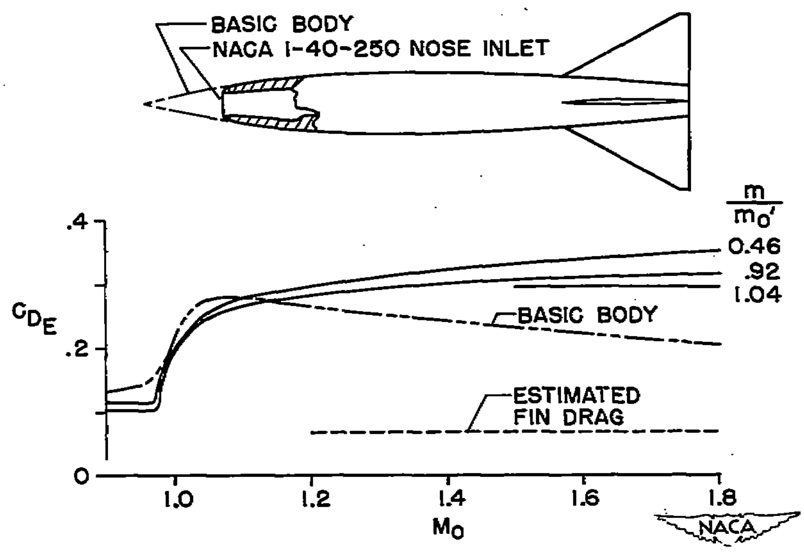

Figure 4.- Drag characteristics of NACA 1-series nose inlet at transonic and low supersonic speeds. $\alpha = 0^{\circ}$, $R = 3.2$ to 10.5×10^{6}.

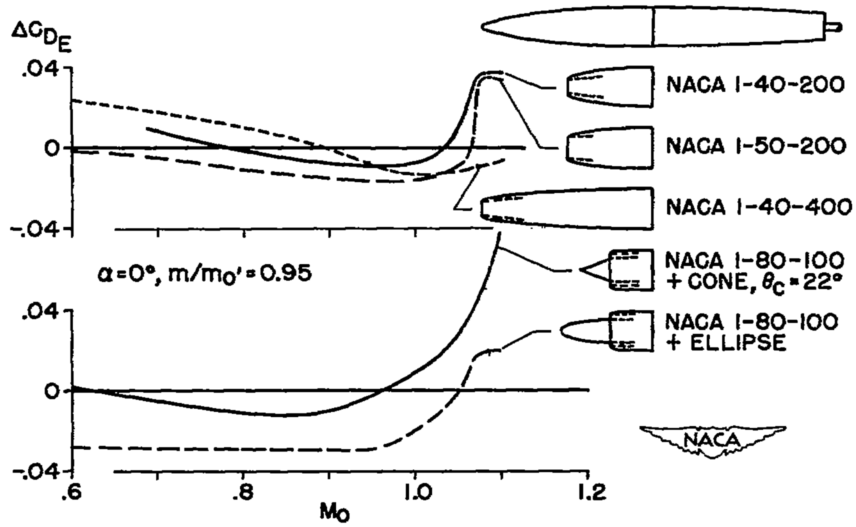

Figure 5.- Effects of inlet proportions and central bodies on transonic drag characteristics of several NACA 1-series nose inlets. $R = 2.3$ to 2.7×10^6.

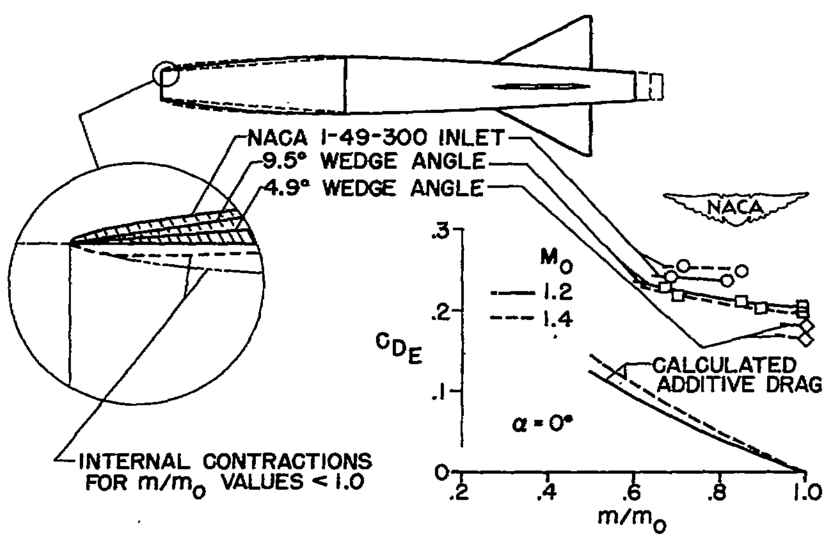

Figure 6.- Effects of lip shape and inlet profile on drag characteristics of open-nose body at low supersonic speeds. $R = 4.7$ to 5.3×10^6.

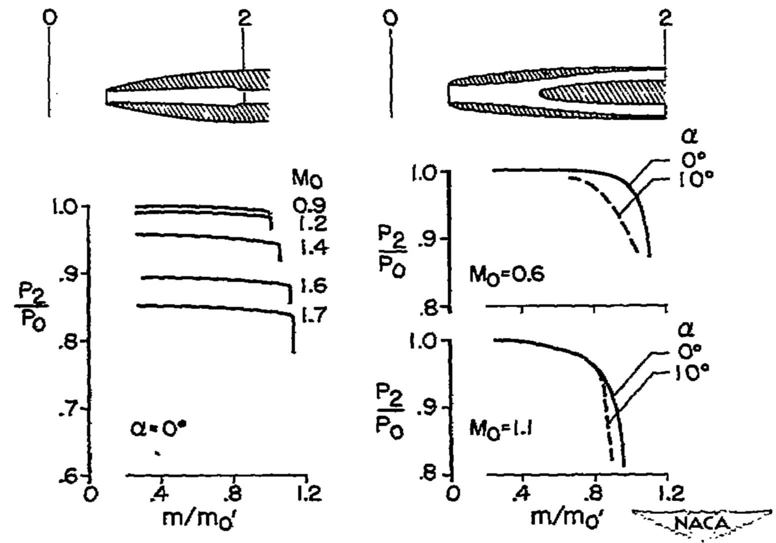

Figure 7.- Pressure-recovery characteristics of two nose inlets at transonic and low supersonic speeds. $R = 4$ to 9×10^6 for inlet on left and 2.3 to 2.7×10^6 for inlet on right.

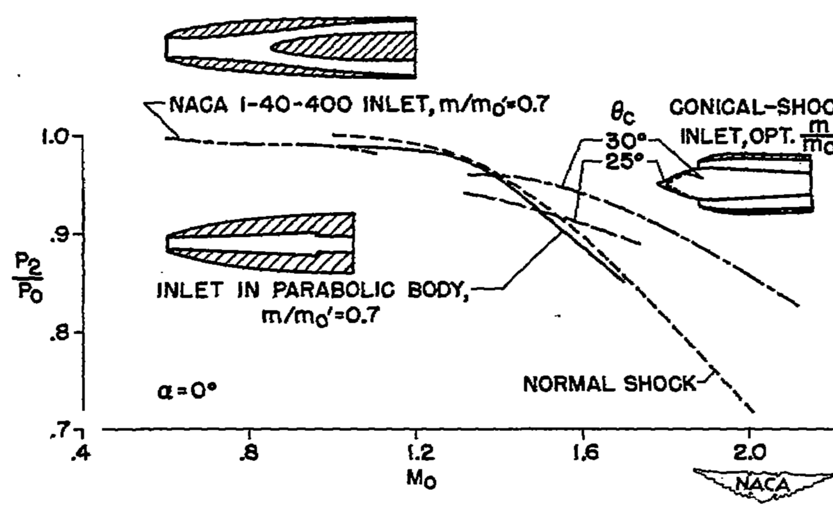

Figure 8.- Comparison of pressure recoveries of open-nose and conical-shock nose inlets. $R = 3.5$ to 4.5×10^6 for conical-shock inlets.

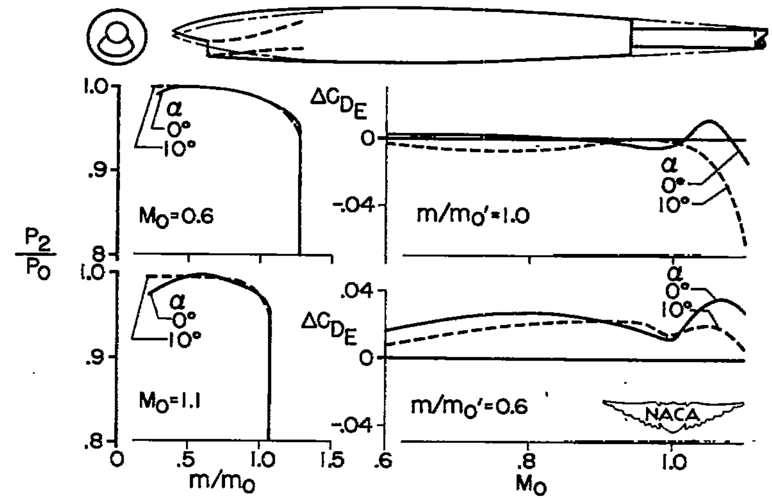

Figure 9.- Transonic pressure-recovery and drag characteristics of forward underslung scoop. $R = 2.3$ to 2.7×10^6.

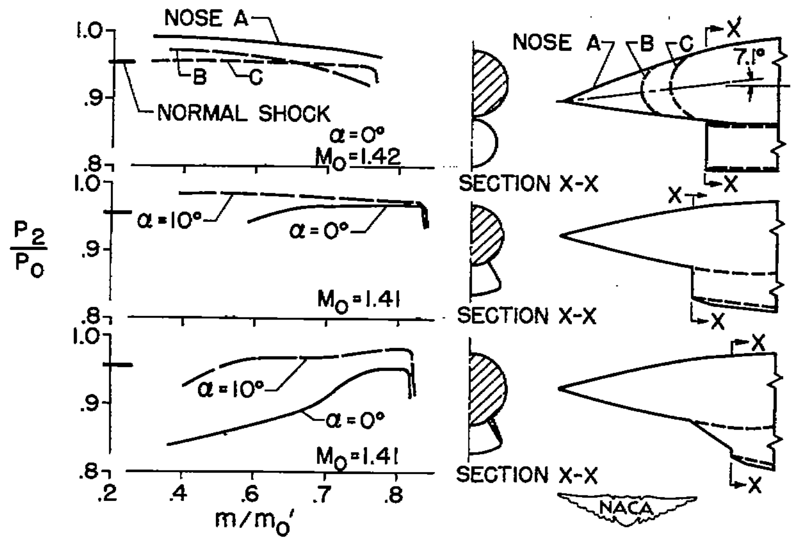

Figure 10.- Supersonic pressure recovery of three forward-underslung scoops. $R = 12.5 \times 10^6$ for inlet at top and 1.3×10^6 for two inlets at bottom.

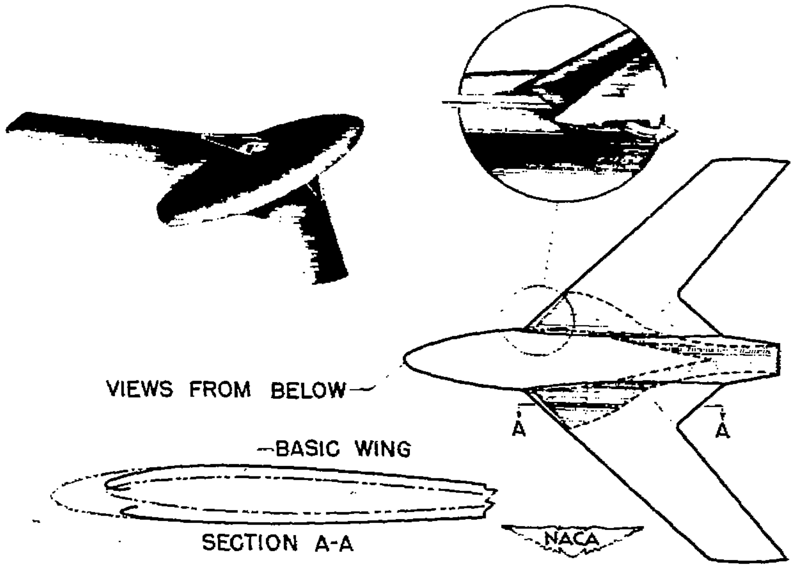

Figure 11.- Swept-wing root inlet.

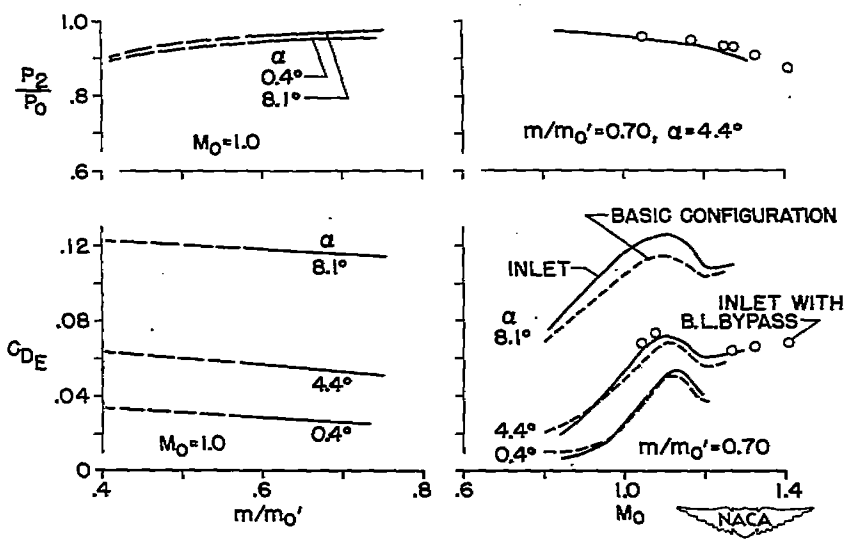

Figure 12.- Transonic pressure-recovery and drag characteristics of swept-wing root inlet. R = 3.7 to 4.9×10^6.

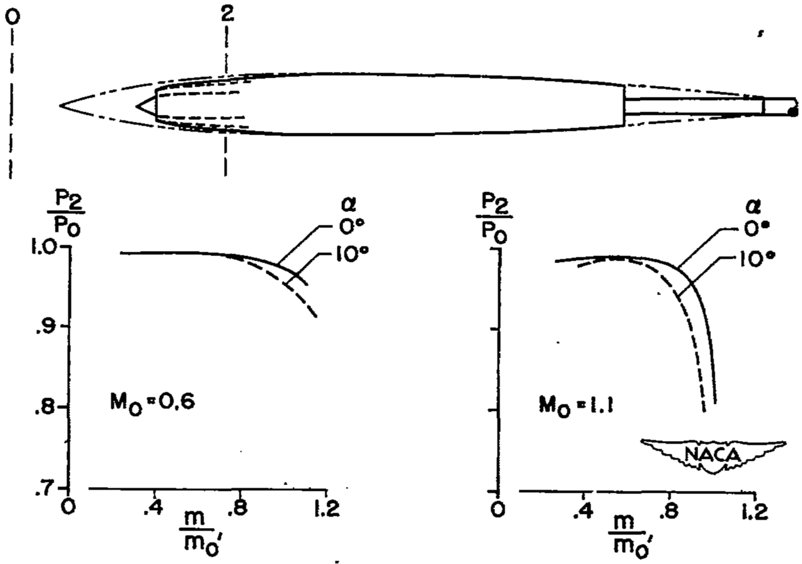

Figure 13.- Transonic pressure recovery of conical-shock supersonic inlet designed for $M_0 = 2.0$. $R = 2.3$ to 2.7×10^6.

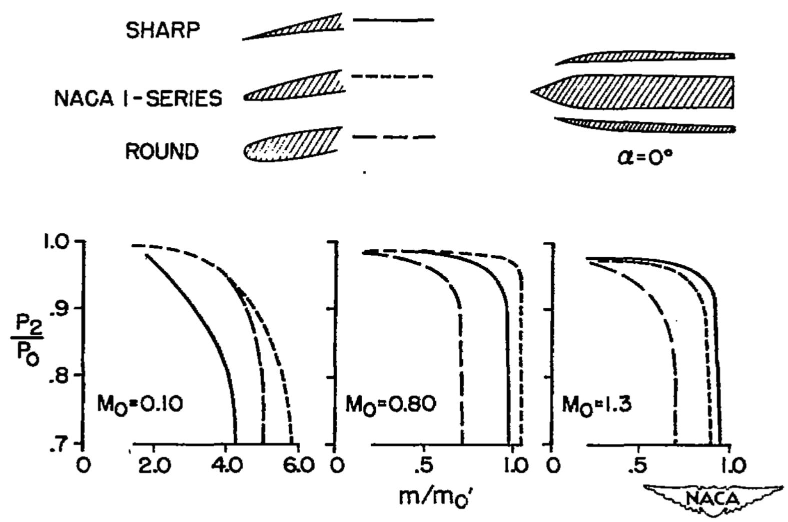

Figure 14.- Effect of lip rounding on pressure recovery of conical-shock supersonic inlet at subsonic and transonic speeds. $R = 0.37$ to 6.9×10^6.

SECURITY INFORMATION

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